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Kupratis

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(54) **GAS TURBINE ENGINE WITH INTERCOOLING TURBINE SECTION**

(75) Inventor: **Daniel B. Kupratis**, Wallingford, CT (US)

(73) Assignee: **United Technologies Corporation**, Hartford, CT (US)

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F02C 9/54	(2006.01)

(52) **U.S. Cl.**

CPC . ***F02C 7/143*** (2013.01); ***F02C 9/20*** (2013.01); *F02C 9/22* (2013.01); *F02C 9/54* (2013.01); ***F02K 3/075*** (2013.01); *Y02T 50/675* (2013.01)

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USPC 60/226.1, 262, 226.3, 39.23, 805, 728

See application file for complete search history.

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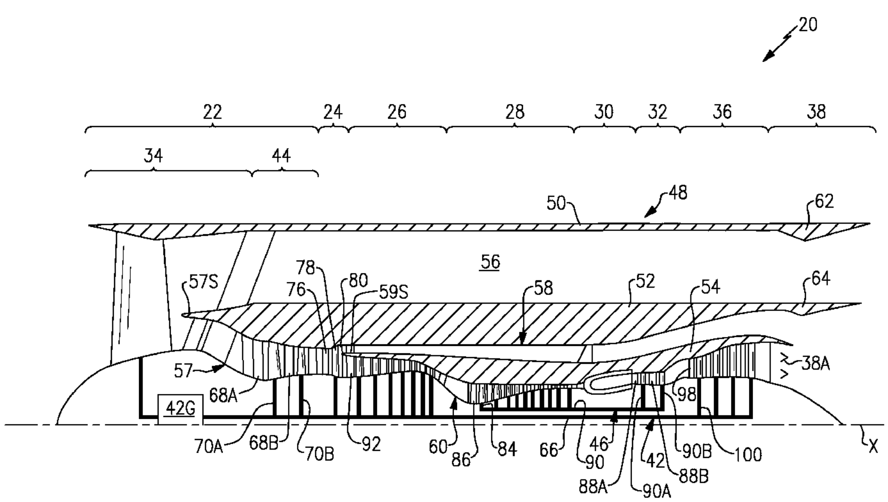
Primary Examiner — William H Rodriguez

(74) *Attorney, Agent, or Firm* — Carlson, Gaskey & Olds, P.C.

(57) **ABSTRACT**

A gas turbine engine includes a combustor section, a fan section forward of the combustor section, a low pressure turbine section along the combustor section, and an intercooling turbine section aft of the fan section and forward of the combustor section. The intercooling turbine section includes upstream and downstream intercooling turbine variable vanes. The intercooling turbine section is situated in an intermediate flow path that is inboard of an outer bypass flow path. The intermediate flow path splits downstream from the intercooling turbine section to a second stream bypass flow path and a core flow path. The second stream bypass flow path is inboard of the outer bypass flow path and extends to an exhaust nozzle. The exhaust nozzle is located aft of the low pressure turbine section and inboard of the outer bypass flow path.

22 Claims, 2 Drawing Sheets



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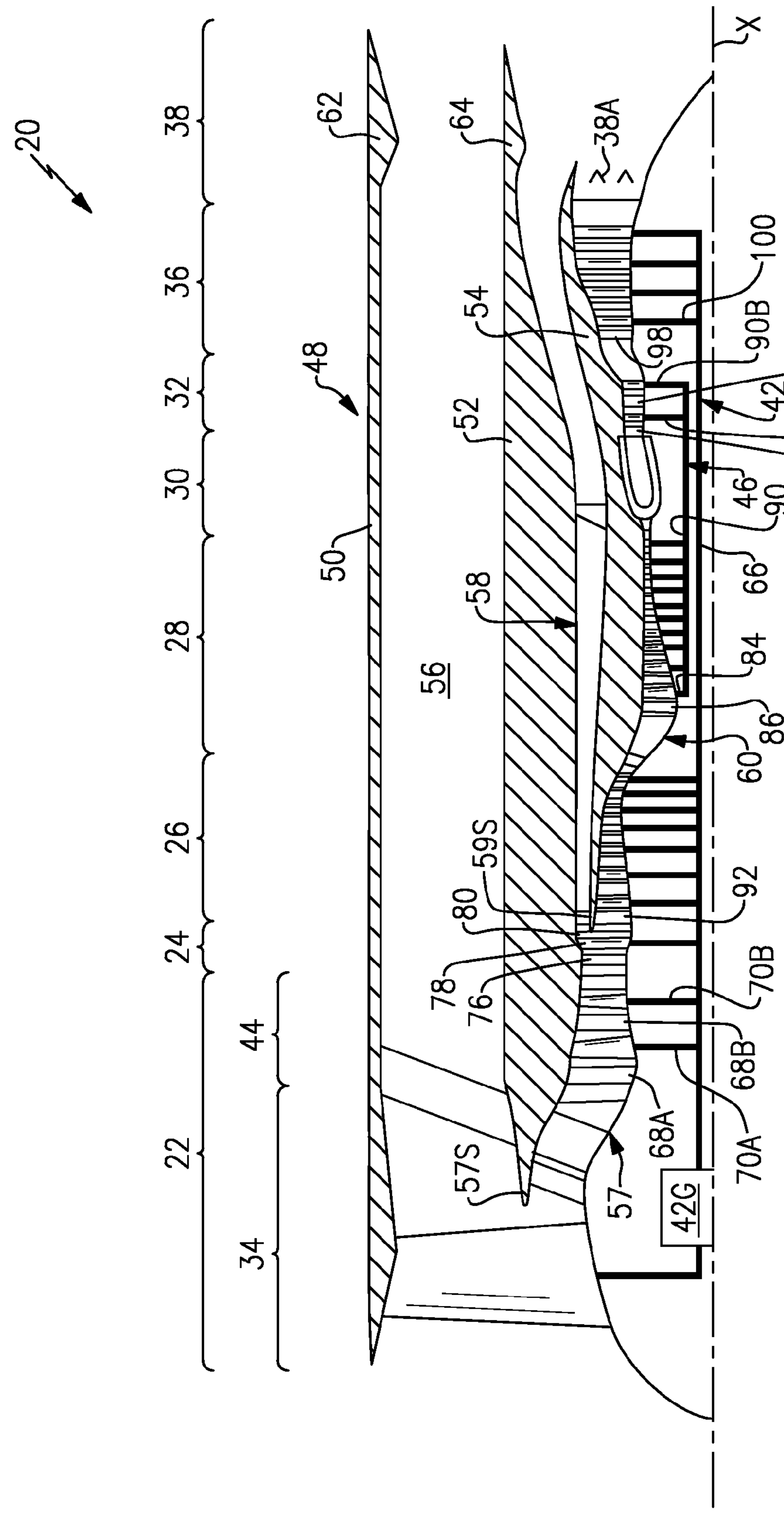
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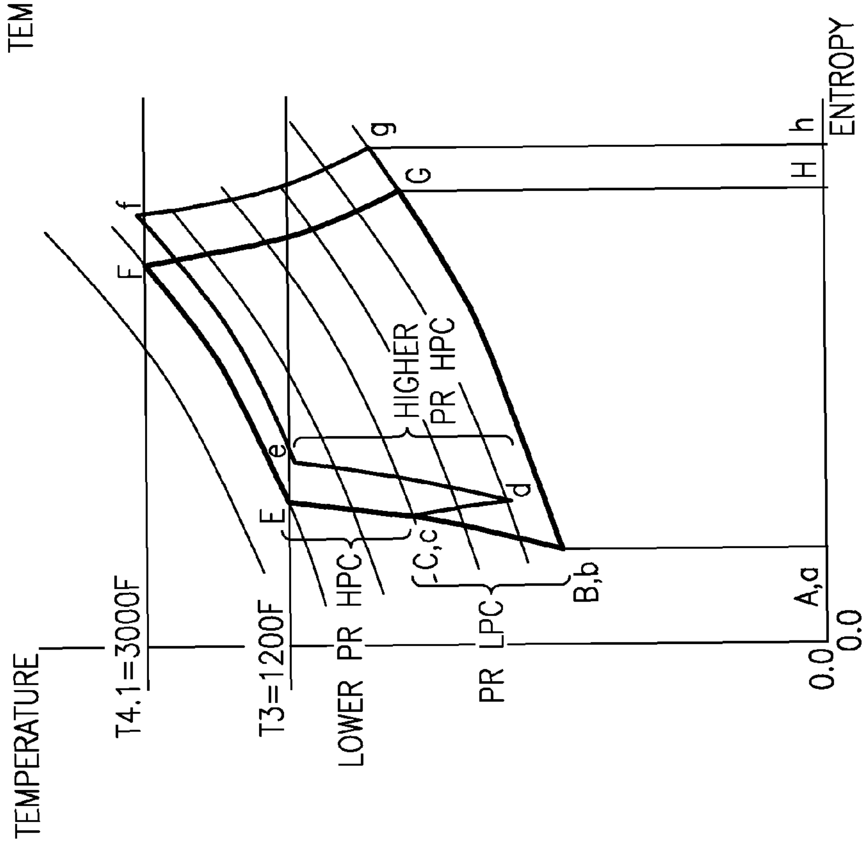
FIG.1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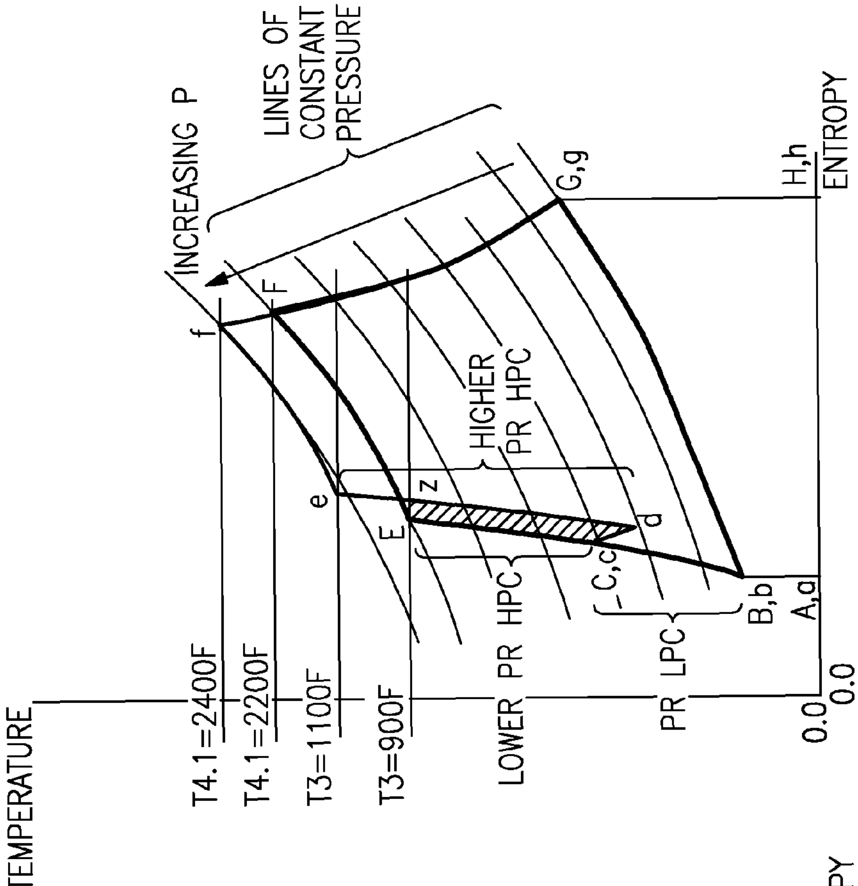
ENTROPY

FIG.2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d
HIGHER PR HPC
F
f
G,g
H,h
ENTROPY
INCREASING P
LINES OF CONSTANT PRESSURE

FIG.3

GAS TURBINE ENGINE WITH INTERCOOLING TURBINE SECTION

BACKGROUND

The present disclosure relates to gas turbine engines, and more particularly to a variable cycle gas turbine engine.

Variable cycle engines power high performance aircraft over a range of operating conditions yet achieve countervailing objectives such as high specific thrust and low fuel consumption. The variable cycle engine essentially alters a bypass ratio during flight to match requirements. This facilitates efficient performance over a broad range of altitudes and flight conditions to generate high thrust when needed for high energy maneuvers yet also optimize fuel efficiency for cruise and loiter conditions.

SUMMARY

A gas turbine engine according to an exemplary aspect of the present disclosure includes a fan section along an engine axis forward of a combustor section. A low pressure turbine section along the engine axis is aft of the combustor section. An intercooling turbine section is along the engine axis aft of the fan section and forward of the combustor section.

A gas turbine engine according to an exemplary aspect of the present disclosure includes a low spool along an engine axis with a fan section and an intercooling turbine section forward of a combustor section. A high spool along the engine axis with a high pressure compressor section and a high pressure turbine section, the high pressure compressor section forward of the combustor section and the high pressure turbine section aft of the combustor section.

A method of operating a gas turbine engine according to an exemplary aspect of the present disclosure includes modulating a guide vane of an intercooling turbine section forward of a combustor section to reduce the intercooling turbine expansion pressure ratio (ICT PR) during a first flight condition; and modulating the guide vane of the intercooling turbine section to increase the intercooling turbine expansion pressure ratio (ICT PR) during a second flight condition.

BRIEF DESCRIPTION OF THE DRAWINGS

Various features will become apparent to those skilled in the art from the following detailed description of the disclosed non-limiting embodiment. The drawings that accompany the detailed description can be briefly described as follows:

FIG. **1** is a general schematic view of an exemplary variable cycle gas turbine engine according to one non-limiting embodiment;

FIG. **2** is a temperature-versus-entropy diagram for a high/hot day take off condition with example temperature distributions; and

FIG. **3** is a temperature-versus-entropy diagram for a cruise condition with example temperature distributions.

DETAILED DESCRIPTION

FIG. **1** schematically illustrates a gas turbine engine **20**. The gas turbine engine **20** is disclosed herein as a variable cycle two-spool bypass turbofan that generally includes a fan section **22**, an intercooling turbine section (ICT) **24**, a low pressure compressor section (LPC) **26**, a high pressure compressor section (HPC) **28**, a combustor section **30**, a high pressure turbine section (HPT) **32**, a low pressure turbine section (LPT) **36**, and a nozzle section **38**. Additional sections may include an augmentor section **38**A, various duct sections, and among other systems or features such as a geared architecture **42**G which may be located in various other engine sections than that shown such as, for example, aft of the LPT. The sections are defined along a central longitudinal engine axis X.

The engine **20** generally includes a low spool **42** and a high spool **46** which rotate about the engine central longitudinal axis X relative to an engine case structure **48**. It should be appreciated that other architectures, such as a three-spool architecture, will also benefit herefrom.

The engine case structure **48** generally includes an outer case structure **50**, an intermediate case structure **52** and an inner case structure **54**. It should be understood that various structures individually or collectively within the engine may define the case structures **50**, **52**, **54** to essentially define an exoskeleton that supports the spools **42**, **46** for rotation therein.

The fan section **22** generally includes a bypass fan **34** and a multistage fan **44**. The low spool **42** drives the bypass fan **34** directly or through a geared architecture **42**G to drive the bypass fan **34** at a lower speed than the low spool **42**. The bypass fan **34** communicates fan flow into a bypass flow path **56**, a second stream bypass flow path **58**, and a core flow path **60**.

The multistage fan **44** in the disclosed non-limiting embodiment, includes two stages downstream of the bypass fan **34**. It should be appreciated that various fan stages may alternatively or additionally be provided. The multistage fan **44** is within an intermediate flow path **57** upstream of a split **59**S between the second stream bypass flow path **58** and the core flow path **60** but downstream of a split **57**S between the bypass flow path **56** and the intermediate flow path **57** such that all airflow from the multistage fan **44** is expanded through the intercooling turbine section **24**. The intercooling turbine section **24** facilitates the expansion of the airflow to a lower temperature than at the exit of the multistage fan **44** and therefore the inlet temperature to the low pressure compressor section **26** is reduced.

The intercooling turbine section **24** communicates fan flow into the second stream bypass flow path **58** and the core flow path **60**. The intercooling turbine section **24** is downstream of the multistage fan **44** such that all flow from the intercooling turbine section **24** is selectively communicated into the second stream bypass flow path **58** and the core flow path **60**. That is, the intercooling turbine section **24** is upstream of the split **59**S between the second stream bypass flow path **58** and the core flow path **60** but downstream of the split **57**S between the bypass flow path **56** and the intermediate flow path **57**.

The low pressure compressor section **26**, the high pressure compressor section **28**, the combustor section **30**, the high pressure turbine section **32**, and the low pressure turbine section **36** are in the core flow path **60**. These sections are referred to herein as the engine core. The core airflow is compressed by the fan section **22**, expanded limitedly by the intercooling turbine section **24**, and compressed monotonically by the low pressure compressor section **26** and the high pressure compressor section **28**, mixed and burned with fuel in the combustor section **30**, then expanded over the high pressure turbine section **32** and the low pressure turbine section **36**. The turbines **32**, **36** rotationally drive the respective high spool **46** and low spool **42** in response to the expansion. The limited expansion of the core flow by the ICT **24** rotationally drives the low spool **42** as a supplement to the LPT **36**.

The second stream bypass flowpath **58** permits the match of the intercooling turbine section **24** exit flow to the flow demand into the low pressure compressor section **26**. That is, the intercooling turbine section **24** expands fan section **22** flow to reduce inlet temperatures to the low pressure compressor section **26**.

The bypass flow path **56** is generally defined by the outer case structure **50** and the intermediate case structure **52**. The second stream bypass flowpath **58** is generally defined by the intermediate case structure **52** and the inner case structure **54**. The core flow path **60** is generally defined by the inner case structure **54**. The second stream bypass flow path **58** is defined radially inward of the bypass flow path **56** and the core flow path **60** is radially inward of the bypass flowpath **58**.

The nozzle section **38** may include a bypass flow exhaust nozzle **62** (illustrated schematically) which receives flow from bypass flow path **56** and a mixed flow exhaust nozzle **64** which receives a mixed flow from the second stream bypass flowpath **58** and the core flow path **60**. It should be understood that various fixed, variable, convergent/divergent, two-dimensional and three-dimensional nozzle systems may be utilized herewith.

The multistage fan **44**, the low pressure compressor section **26**, the intercooling turbine section **24**, and the low pressure turbine section **36** are coupled by a low shaft **66** which is also coupled to the bypass fan **34** directly or through the geared architecture **42**G. In the disclosed non-limiting embodiment, the multistage fan **44** includes a first stage guide vane **68**A, a first stage fan rotor **70**A, a second stage guide vane **68**B and a second stage fan rotor **70**B. It should be appreciated that various systems may be utilized to activate the variable inlet guide vanes and variable stators. It should also be understood that other single or multistage architectures may alternatively or additionally be provided such as various combinations of a fixed or variable vanes.

The high pressure compressor section **28** and the high pressure turbine section **32** are coupled by a high shaft **90** to define the high spool **46**. In the disclosed non-limiting embodiment, the high pressure compressor section **28** upstream of the combustor section **30** includes a multiple of stages each with a rotor **84** and vane **86**. It should be understood that the high pressure compressor section **28** may alternatively or additionally include other compressor section architectures which, for example, include additional or fewer stages each with or without various combinations of variable or fixed guide vanes. It should also be understood that each of the turbine sections **32**, **36** may alternatively or additionally include other turbine architectures which, for example, include additional or fewer stages each with or without various combinations of variable or fixed guide vanes.

The high pressure turbine section **32** in the disclosed non-limiting embodiment, includes a multiple of stages (two shown) with first stage variable high pressure turbine guide vanes **88**A, a first stage high pressure turbine rotor **90**A, second stage variable high pressure turbine guide vanes **88**B and a second stage high pressure turbine rotor **90**B. It is desirable to have variable turbine vanes both in the intercooling turbine section **24** and the high pressure turbine section **32** to facilitate an optimum cycle efficiency; however, significant improvement to the thermodynamic cycle is achieved by the presence of the intercooling turbine section **24** alone even if the high pressure turbine section **32** is not variable.

The low pressure turbine section **36** in the disclosed non-limiting embodiment, includes a multiple of stages (four shown), each stage with variable low pressure turbine inlet guide vanes **98** upstream of a low pressure turbine rotor **100**. It is desirable to have variable turbine vanes both in the intercooling turbine section **24** and the low pressure turbine section **36** to facilitate an optimum cycle efficiency; however, significant improvement to the thermodynamic cycle is achieved by the presence of the intercooling turbine section **24** alone even if the low pressure turbine section **36** is not variable. The low pressure turbine section **36** is the last turbine section within the core flow path **60** and thereby communicates with the mixed flow exhaust nozzle **64** which receives a mixed flow from the second stream bypass duct **58** and the core flow path **60**. The augmentor section **38**A among other systems or features may be located downstream of the low pressure turbine section **36**.

The intercooling turbine section **24** is coupled to the low shaft **66** and drives the low spool **42**. In the disclosed non-limiting embodiment, the intercooling turbine section **24** includes upstream intercooling turbine variable vanes **76**, an intercooling turbine rotor **78** and downstream intercooling turbine variable vanes **80**. The downstream intercooling turbine variable vanes **80** is immediately upstream of the split **59**S between the second stream bypass flow path **58** and the core flow path **60**. It should be appreciated that the intercooling turbine rotor **78** is a cold turbine located forward of the combustor section **30** but includes turbine blades similar in shape to the turbine blades within the high pressure turbine section **32** and the low pressure turbine section **36**.

In the disclosed non-limiting embodiment, the low pressure compressor section **26** includes a multiple of stages (seven shown) with inlet guide vanes (IGVs) **92** immediately downstream of the downstream intercooling turbine variable vanes **80**. The LPC IGVs **92** are within the core flow path **60** downstream of the split **59**S. At cruise, the IGVs **92** are closed, and at takeoff, the IGVs **92** are opened. These IGV settings are similar in function to the downstream intercooling turbine variable vanes **80** such that the downstream intercooling turbine variable vanes **80** may be eliminated in one alternative non-limiting embodiment to simplify the engine architecture and in which the LPC IGVs **92** are relied upon alone to perform a common function for both the intercooling turbine section **24** and the LPC **26**.

The high spool **46** is independent of the low spool **42**. The high pressure compressor section **28**, the combustor section **30** and the high pressure turbine section **32** set the core flow and the high pressure turbine section **32** is the choke point that the high pressure compressor section **28** feeds. The speed and power output of the high pressure turbine section **32** determines the flow pumping rate and the pressure rise of the high pressure compressor section **28** for a given combustor exit temperature that affects the status of the choke point.

The high pressure compressor section **28** demands a certain inlet flow based on combustor exit temperature. There is a balance between the pressure at the exit of the high pressure compressor section **28** and the temperature at the choke point. Higher combustor section exit temperature enables higher pressure at the exit of the high pressure compressor section **28** and a simultaneous solution of continuity (flowrate) and power from the high pressure turbine section **32**.

The low pressure compressor section **26** must pass the flow required by the high pressure compressor section **28** but this may result in a matching of the low pressure compressor section **26** that stalls the low pressure compressor section **26**. The flow which exits the intercooling turbine section **24** is split between the low pressure compressor section **26** and the second stream bypass flowpath **58** around the low pressure compressor section **26** which facilitates the ability of the low pressure compressor section **26** to provide a matched flow to the low pressure compressor section **26** that is not a stalled condition for the low pressure compressor section **26** itself.

Opening the upstream intercooling turbine variable vanes **76** and closing the downstream intercooling turbine variable vanes **80** for cruise reduces the intercooling turbine section pressure ratio (ICT PR) and hence reduces the intercooling effect, e.g., the inlet temperature to the LPC **26** will not be significantly decreased. Closing the upstream intercooling turbine variable vanes **76** and opening the downstream intercooling turbine variable vanes **80** for takeoff will increase ICT PR and hence increase the intercooling effect, e.g., the inlet temperature to the LPC **26** will be more significantly decreased. In the disclosed non-limiting embodiment, the intercooling turbine variable vanes **76**, **80** are modulated between a 5%-25% closed position. That is, the intercooling turbine variable vanes **76**, **80** are never fully closed.

Generally, the intercooling variable vanes **80** are opened for takeoff to increase the pressure ratio and intercooling effect to reduce combustor inlet temperature (T**3**) on hot day conditions (FIG. **2**). The intercooling variable vanes **80** are closed for cruise to reduce the intercooling turbine expansion pressure ratio (ICT PR) and the intercooling effect (FIG. **3**).

The second stream bypass flowpath **58** permits the match of the intercooling turbine section **24** exit flow to the flow demand into the low pressure compressor section **26**. That is, the intercooling turbine section **24** expands fan section **22** flow to reduce inlet temperatures to the low pressure compressor section **26**.

With reference to FIGS. **2** and **3**, a conventional engine cycle is defined thermodynamically on an example Temperature-Entropy diagram by the points A, B, C, E, F, G, H. The priority for improvement of the thermodynamic efficiency of the engine is to increase the area enclosed by the points B, C, E, F, G, but especially doing so by "raising the roof" of points (E) and (F) that correspond respectively to an increase in the overall PR of the engine compression system (E) and an increase in the inlet temperature to the HPT **32** (F). It should be appreciated that the temperatures are merely exemplary to one disclosed non-limiting embodiment and should not be considered otherwise limiting.

The inventive engine cycle disclosed herein is defined thermodynamically on the Temperature-Entropy diagram by points a, b, c, d, e, f, g, h. The priority is improvement of the cruise condition efficiency where significant fuel is consumed.

Both the conventional engine and the inventive engine **20** architectures disclosed herein operate at the hot day takeoff condition (FIG. **2**) with the same inlet temperature and pressure to the engine, the same inlet temperature and pressure to the multistage fan **44**: TB=Tb; and PB=Pb, as well as the same temperature and pressure at the exit of the multistage fan **44**: TC=Tc; and PC=Pc.

For the conventional engine, the inlet temperature and pressure to the HPC **28** is the same as the exit temperature and pressure of the LPC **26**, however, for the inventive engine disclosed herein the inlet temperature and pressure to the HPC **28** are Td and Pd, respectively. The ICT **24** expands the exit flow of the multistage fan **44** so that the inlet temperature and pressure to the HPC **28** of the inventive engine are decreased significantly to achieve an intercooling effect on the temperature of compression, that is, Td<TC and Pd<PC.

For both the conventional engine and the inventive engine, the exit condition of the HPC **28** is the inlet condition of the combustor section **30**. Both the conventional engine and the inventive engine operate at the hot day takeoff condition with the same combustor inlet temperature (T**3**), where TE=Te, and with the same HPT **32** first rotor inlet temperature, (T**4.1**), where TF=Tf. This is consistent with utilization of the same materials and mechanical design technologies for both the conventional and inventive engine.

The pressure ratio (PR) of the HPC **28** of the inventive engine is significantly higher than the PR of the conventional engine, that is, Pe:Pd>PE:PC. The temperature ratio (TR) of the HPC **28** of the inventive engine is significantly higher than the TR of the conventional engine, that is, Te:Td>TE:TC. The higher PR of the HPC **28** of the inventive engine **20** is achievable, for example, with additional compressor section stages.

Neglecting combustor pressure losses, the pressures, PE and PF for the conventional engine are the same. The pressures, Pe and Pf, for the inventive engine are the same, but PE>Pe and PF>Pf; this is attributable to the pressure expansion in the ICT **24**.

Both the conventional engine and the inventive engine operate with the same HPT **32** first rotor inlet temperature (T**4.1**), and TF=Tf at the hot day takeoff condition. At the hot day takeoff condition, both the conventional engine and the inventive engine operate with the same exit pressure from the turbine section so that PG=Pg, but not the same exit temperature from the turbine section, that is, Tg>TG.

The thermodynamic cycle efficiency of an engine generally is proportional to the ratio of two areas on the Temperature-Entropy diagram. That is, the numerator area and the denominator area form this ratio of areas. For the conventional engine, the numerator area is enclosed by the points B, C, E, F, and G, while the denominator area is enclosed by the points H, G, B, and A. For the inventive engine, the numerator area is enclosed by the points b, c, d, e, f, and g, while the denominator area is enclosed by the points h, g, b, and a.

At the hot day takeoff condition, the numerator area of the conventional engine is greater than or equal to the numerator area of the inventive engine, while the denominator area of the conventional engine is less than the denominator area of the inventive engine; thus, the thermodynamic efficiency of the conventional engine is relatively better than the inventive engine at the hot day takeoff condition (FIG. **2**).

The priority, however, is to improve the thermodynamic cycle efficiency at the cruise condition where much of the fuel is consumed. Both the conventional engine and the inventive engine operate at the cruise condition with the same inlet temperature and pressure to the engine and the same inlet temperature and pressure to the multistage fan **44**; TB=Tb and PB=Pb. Note that the inlet temperature and pressure at the cruise condition (FIG. **3**) are less than the inlet temperature and pressure at the hot day takeoff condition (FIG. **2**).

At the cruise condition, both the conventional and inventive engine have the same temperature and pressure at the exit of the multistage fan **44**; TC=Tc and PC=Pc. For the conventional engine, the inlet temperature and pressure to the HPC **28** is the same as the exit temperature and pressure of the LPC **26**; however, for the inventive engine **20**, the inlet temperature and pressure to the HPC **28** are Td and Pd, respectively. At the cruise condition, the ICT **24** expands the exit flow of the multistage fan **44** so that the inlet temperature and pressure to the HPC **28** of the inventive engine are not decreased significantly to obtain a smaller intercooling effect on the temperature within the compressor section; regardless, Td<TC and Pd<PC.

The expansion of the ICT **24** is selectively less at the cruise condition and this is obtained by modulation of the variable vanes **76** and **80**. At the cruise condition as well as the hot day takeoff condition, the HPC **28** of the inventive engine **20** has a higher PR than the conventional engine and the higher PR is achieved for example, with additional stages of compression in the HPC **28**. The pressure ratio (PR) of the HPC **28** of the inventive engine is significantly higher than the PR of the conventional engine, that is, Pe:Pd>PE:PC. The temperature ratio (TR) of the HPC of the inventive engine is significantly higher than the TR of the conventional engine, that is, Te:Td>TE:TC. At the cruise condition, the HPC **28** exit pressure and exit temperature of the inventive engine are higher than the conventional engine, that is, Pe>PE and Te>TE.

For both the conventional engine and the inventive engine, the exit condition of the HPC **28** is the inlet condition of the combustor section **30**. Neglecting combustor pressure losses, the pressures, PE and PF for the conventional engine are the same. The pressures, Pe and Pf for the inventive engine are the same, but at the cruise condition, Pf>PF; this is attributed to the deliberately smaller expansion of pressure in the ICT **24** and the higher PR of the HPC **28** of the inventive engine.

Application of the same materials and mechanical design technologies to both the conventional and inventive engine is limiting at the hot day takeoff condition but not at the cruise condition provided T**3** and T**4.1** at the cruise condition are lower than at the hot day takeoff condition.

At the cruise condition, HPT **32** first rotor inlet temperature (T**4.1**) of the inventive engine is greater than T**4.1** of the conventional engine; that is, Tf>TF at the cruise condition. At the cruise condition, both the conventional engine and the inventive engine operate with the same exit pressure of the turbine section so that PG=Pg, and the same exit temperature from the turbine section, TG=Tg.

With reference to FIG. **3**, at the cruise condition, the numerator area of the inventive engine is greater than the numerator area of the conventional engine, while the denominator areas of the conventional engine and the inventive engine are the same; thus, the thermodynamic efficiency of the inventive engine is greater than the conventional engine at the cruise condition. The larger numerator area of the inventive engine is evident by comparison between the two sectional areas of the Temperature-Entropy diagram at the cruise condition.

The first sectional area is enclosed by the points z, e, f, and F, while the second sectional area is enclosed by the points C, E, z, and d. The first sectional area yields an increase in the numerator area of the inventive engine while the second sectional area yields a reduction in the numerator area of the inventive engine. The first sectional area is greater than the second sectional area to yield a net increase in the numerator area of the inventive engine disclosed herein versus the numerator area of the conventional engine.

The ICT **24** effectively "raises the roof" of the thermodynamic cycle of the engine at the cruise condition with the same materials and mechanical design constraints as a conventional engine architecture at the hot day takeoff condition.

It should be understood that relative positional terms such as "forward," "aft," "upper," "lower," "above," "below," and the like are with reference to the engine but should not be considered otherwise limiting.

Although the different non-limiting embodiments have specific illustrated components, the embodiments of this invention are not limited to those particular combinations. It is possible to use some of the components or features from any of the non-limiting embodiments in combination with features or components from any of the other non-limiting embodiments.

It should be understood that like reference numerals identify corresponding or similar elements throughout the several drawings. It should also be understood that although a particular component arrangement is disclosed in the illustrated embodiment, other arrangements will benefit herefrom.

Although particular step sequences are shown, described, and claimed, it should be understood that steps may be performed in any order, separated or combined unless otherwise indicated and will still benefit from the present disclosure.

The foregoing description is exemplary rather than defined by the limitations within. Various non-limiting embodiments are disclosed herein, however, one of ordinary skill in the art would recognize that various modifications and variations in light of the above teachings will fall within the scope of the appended claims. It is therefore to be understood that within the scope of the appended claims, the disclosure may be practiced other than as specifically described. For that reason the appended claims should be studied to determine true scope and content.

What is claimed is:

1. A gas turbine engine comprising:

a combustor section;

a fan section along an engine axis forward of said combustor section;

a low pressure turbine section along said engine axis aft of said combustor section; and

an intercooling turbine section along said engine axis aft of said fan section and forward of said combustor section, said intercooling turbine section including an upstream intercooling turbine variable vane, an intercooling turbine rotor, and a downstream intercooling turbine variable vane, and

wherein said intercooling turbine section is situated in an intermediate flow path that is inboard of an outer bypass flow path, said intermediate flow path splitting downstream from said intercooling turbine section into a second stream bypass flow path and a core flow path, said second stream bypass flow path being inboard of said outer bypass flow path and extending to an exhaust nozzle, said exhaust nozzle being located aft of said low pressure turbine section and inboard of said outer bypass flow path.

2. The gas turbine engine as recited in claim **1**, wherein said fan section includes a bypass fan and a multistage fan.

3. The gas turbine engine as recited in claim **1**, further comprising a low pressure compressor section downstream of said intercooling turbine section.

4. The gas turbine engine as recited in claim **1**, further comprising a high pressure compressor section downstream of said intercooling turbine section and upstream of said combustor section.

5. The gas turbine engine as recited in claim **1**, further comprising a high pressure turbine section downstream of said combustor section and upstream of said low pressure turbine section.

6. The gas turbine engine as recited in claim **1**, wherein said downstream intercooling turbine variable vane is immediately upstream of said split in said intermediate flow path between said second stream bypass flow path and said core flow path.

7. The gas turbine engine as recited in claim **6**, wherein said combustor section is in communication with said core flow path.

8. The gas turbine engine as recited in claim **6**, further comprising a high pressure compressor section and a high pressure turbine section in communication with said core flow path.

9. The gas turbine engine as recited in claim **8**, further comprising a low pressure compressor section downstream of said intercooling turbine section in communication with said core flow path.

10. The gas turbine engine as recited in claim **9**, further comprising an inlet guide vane upstream of said low pressure compressor section and downstream of said intercooling turbine section, said inlet guide vane within said core flow path.

11. The gas turbine engine as recited in claim **1**, wherein said upstream intercooling turbine variable vane and said downstream upstream intercooling turbine variable vane are configured to modulate intercooling turbine expansion pressure ratio responsive to first and second flight conditions.

12. A gas turbine engine comprising:

a combustor section;

a low spool along an engine axis with a fan section and an intercooling turbine section forward of said combustor section; and

a high spool along said engine axis with a high pressure compressor section and a high pressure turbine section, said high pressure compressor section forward of said combustor section and said high pressure turbine section aft of said combustor section, and

wherein said intercooling turbine section includes upstream and downstream intercooling turbine variable vanes configured to modulate intercooling turbine expansion pressure ratio responsive to first and second flight conditions.

13. The gas turbine engine as recited in claim **12**, wherein low spool includes a low pressure compressor section aft of said intercooling turbine section and forward of said combustor section.

14. The gas turbine engine as recited in claim **12**, wherein low spool includes a low pressure turbine section aft of said combustor section.

15. The gas turbine engine as recited in claim **12**, wherein said fan section includes a bypass fan and a multistage fan, said bypass fan driven by said low spool through a geared architecture.

16. The gas turbine engine as recited in claim **15**, wherein said bypass fan communicates with a bypass flow path generally defined by the outer case structure and an intermediate case structure, a second stream bypass flowpath generally defined by said intermediate case structure and an inner case structure, a core flow path defined by said inner case structure such that said second stream bypass flow path is radially inward of said bypass flow path and said core flow path is radially inward of said bypass flowpath.

17. The gas turbine engine as recited in claim **12**, wherein said first flight condition is a takeoff flight condition and said second flight condition is a cruise flight condition.

18. The gas turbine engine as recited in claim **12**, wherein said intercooling turbine section is situated in an intermediate flow path that is inboard of an outer bypass flow path, said intermediate flow path splitting downstream from said intercooling turbine section into a second stream bypass flow path and a core flow path, said second stream bypass flow path being inboard of said outer bypass flow path and extending to an exhaust nozzle, said exhaust nozzle being located aft of said high pressure turbine section and inboard of said outer bypass flow path.

19. A method of operating a gas turbine engine comprising:

modulating a guide vane of an intercooling turbine section forward of a combustor section to reduce the intercooling turbine expansion pressure ratio (ICT PR) during a first flight condition by closing an upstream intercooling turbine variable vane and opening a downstream intercooling turbine variable vane; and

modulating the guide vane of the intercooling turbine section to increase the intercooling turbine expansion pressure ratio (ICT PR) during a second flight condition by opening the upstream intercooling turbine variable vane and closing the downstream intercooling turbine variable vane.

20. The method as recited in claim **19**, wherein the first flight condition is a takeoff flight condition.

21. The method as recited in claim **19**, wherein the second flight condition is a cruise flight condition.

22. The method as recited in claim **19**, wherein modulating the guide vane includes moving the guide vane between a 5%-25% closed position.

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